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SPACE-VEHICLE ATTITUDE CONTROL

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SPACE VEHICLE ATTITUDE CONTROL *

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ABSTRACT

For the midcourse guidance of a spacecraft, it is essential that the vehicle contain an angular reference and control system in order to execute the maneuver transmitted via the radio link. Also, when radio techniques are employed for navigation or communication in spacecraft, large savings in power may be obtained by utilizing directional antennae (or conversely, for a given amount of available power, longer ranges or wider bandwidths may be used). These functions of orienting the thrust vector of a midcourse rocket motor and of directing an antenna toward the Earth are performed by the attitude-control system in a space vehicle. Other functions performed by the attitude-control system may include orienting solar cells toward the Sun, moving and pointing equipment for scientific experiments, and controlling the temperature of the spacecraft.

The attitude-control system of a space vehicle utilizes sensors to determine variations of attitude from a reference frame and actuators to correct for any deviations which may exist. Typical attitude-control sensors include the following

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items: Sun finder, Sun sensor, planet sensor, horizon seeker, star sensor, and gyroscope. These devices may employ optical, infrared, or inertial techniques. Typical actuators employ mass expulsion devices, momentum interchange devices, or solar pressure. The sensors and actuators must be considered for two different modes of operation. The first is the cruise mode, where the primary function of the vehicle is to travel from one place to another under its own momentum and gravitational forces. The second is the active mode, where the primary function of the vehicle may be to execute a maneuver to correct its trajectory, or to conduct scientific experiments in the vicinity of the Moon or a planet. Other considerations pertaining to space-vehicle attitude control include choice of reference directions, acquisition of reference, and control of the entire vehicle versus orientation of individual devices.

A typical space-vehicle attitude control system is one being developed by the Jet Propulsion Laboratory of the California Institute of Technology for its Ranger spacecraft. The mission of the Ranger spacecraft is to perform certain scientific experiments in the vicinity of the Moon and to deposit a capsule on the surface of the Moon. The Ranger spacecraft utilizes Sun sensors and an Earth sensor to provide references for three-axis stabilization during the cruise phase. During the midcourse maneuver, three single-degree-of-freedom floated gyroscopes are used to stabilize the vehicle in an arbitrary direction with respect to the Sun-Earth celestial references mentioned above. A high-gain communications antenna is so mounted that it is movable with respect to the vehicle, and it is servo-controlled so as to continuously point at the Earth. In addition, a second omnidirectional antenna is employed. The Ranger actuation scheme is a cold-gas system pressurized on the ground prior to the flight. It is an "on-off" system using nitrogen gas released through nozzles by electrically operated valves. During the cruise phase, a limit cycle type of operation is employed to conserve gas.

I. INTRODUCTION

This paper is concerned with the problem of controlling the attitude, or orientation, of a space vehicle during its journey from the Earth to the Moon. The reasons why it may be desired to control the angular orientation of such a space vehicle are most conveniently listed in terms of the functions which require prescribed angular orientations. Table 1 presents such a listing, with the required accuracies, for a typical lunar mission in which the space vehicle must pass within 50 mi of the Moon and take television pictures of the lunar surface.

The simple expedient of spinning the space vehicle can be used to provide stability about two axes if motions of precession can be avoided. This may satisfy many of the requirements listed in Table 1. For example, if the trajectory is such that the spin axis points toward Earth, this can be used to orient a directional antenna. Pictures of the lunar surface can be obtained from a spinning vehicle either by synchronizing the camera with the spin rate, or perhaps by using the spin rate to obtain a scanning motion. Also, power can be obtained by arranging solar cells about the entire surface of the vehicle. However, it is very unlikely that a meaningful correction to the trajectory could be accomplished with a spinning vehicle. Thus, for more advanced missions, in which post-injection trajectory corrections are used and in which several of the above requirements must be satisfied simultaneously, the vehicle will almost certainly employ three-axis attitude stabilization.

It is convenient to divide the problem of attitude control into two parts: (1) sensing, which refers to the process of identifying and acquiring certain reference directions and then providing signals proportional to variations of the vehicle attitude from this reference frame; and (2) actuation, which refers to the process of applying torques to correct for any deviations indicated by the sensors between the actual attitude and the desired attitude. These two aspects of attitude control will be considered in more detail after some of the more general problems of space-vehicle attitude control have been presented. It is interesting to note that the system comprised of the sensors, the actuators, and the electronic circuitry linking these two types of devices, corresponds to what is ordinarily called an "autopilot" in conventional aircraft or missiles.

Whenever a specific mission is considered, it is usually found that simultaneous, conflicting requirements exist for attitude control. For example, it may be required that, simultaneously, a communications antenna be directed toward the Earth, a camera be pointed toward the Moon, and solar cells be oriented toward the Sun. If all the devices are rigidly attached to the spacecraft, it may not be possible to satisfy simultaneous attitude requirements for the various devices. It then becomes necessary to mount some of the devices on gimballed platforms or swinging arms. The number, location, and allowable motions of such "movable joints" should be treated as an intimate part of the over-all attitude-control problem. Not only must the trade-offs involved in comparing movable-versus-fixed mounting be carefully considered, but, if a movable mounting is selected, its interaction with the rest of the system must also be investigated. Examples of interactions include the fact that any torque exerted on a movable platform will produce an equal and opposite torque on the vehicle, and the fact that the field of view requirements of all devices must be simultaneously satisfied.

In addition to the "mechanization division" made with respect to the functions of sensing and actuation, it is also convenient to divide the attitude-control problem into two categories corresponding to two different modes of operation which occur during the flight. The first mode will be called the "cruise mode," which refers to passive periods of flight when the primary function of the vehicle is to travel from one place to another under its own momentum. The second mode will be called the "active mode," which refers to such periods of flight as the initial acquisition phase (when the attitude references are acquired), the midcourse-maneuver phase (when the trajectory is corrected by firing a rocket motor in the appropriate direction), and the terminal phase (when scientific experiments are conducted in the vicinity of the Moon).

The requirements imposed on the attitude-control system may be considerably different during these two modes of operation. During the active mode, it may be necessary to employ both gyroscopic and celestial references, whereas, the use of only celestial references may be adequate for the cruise mode. As an extreme example, it may be possible to eliminate all attitude control during portions of the cruise mode, and to re-acquire the attitude references later in the flight. Conversely, when a rocket motor is being fired for the midcourse maneuver, the requirements for controlling the direction of the thrust vector are vastly different from the requirements associated with periods of zero thrust. Auxiliary equipment must then be used to accomplish the desired thrust vector control, and, during this midcourse-maneuver period, the resulting attitude-control system may be very similar to a conventional missile autopilot.

A final general consideration, which applies to all space missions within the solar system, is concerned with the use of the Sun as an attitude reference. The Sun is prominent and easy to acquire; therefore, it is very probable that a sensor will be pointed toward the center of the Sun to provide control about two of the axes for any space vehicle employing three-axis attitude control. The probable additional requirements of orienting solar cells toward the Sun to provide electrical power and of utilizing the Sun's radiation for temperature control of the space vehicle enforce the above contention.

Typical sensors and actuators are described in Sections II and III of this paper. In the final Section, a specific attitude-control system being developed for the Ranger spacecraft by the Jet Propulsion Laboratory of the California Institute of Technology is described. It should be emphasized that, although the Ranger attitude-control system is described here by a single author, the actual work was accomplished by many persons in the Guidance and Control Division of the Jet Propulsion Laboratory. Specific credits have been omitted only because of their great number.

II. SENSORS

Some sensors applicable to space-vehicle attitude control are listed in Table 2, together with their application and accuracies. Only those items applicable to the Ranger spacecraft will be considered in detail. It should be noted that, although desired accuracies of 0.001 deg (3.6 sec of arc) are indicated in Table 2, this does not imply that the entire spacecraft will be controlled in attitude to this order of accuracy. It is unlikely that an entire spacecraft would ever be controlled to better than 0.05 deg. Bending of one portion of the spacecraft with respect to other parts can easily cause angular deviations on the order of 0.05 deg (approximately 1 mr). The high accuracies are required in self-contained guidance systems for such functions as determining the angle between the directions toward a star and toward the center of a planet. Such angular measurements can be made with sextant-type instruments which are independent of the exact orientation of the vehicle. However, since the guidance and control functions will be served by the same sensors, the accurate numbers have been included in the Table. The Ranger vehicle utilizes Earth-based radio guidance, rather than self-contained guidance, and all of the accuracies in the "Ranger" column reflect attitude-control requirements.

A. Sun Sensors

The Sun sensors used on the Ranger spacecraft are a simple type which do not require any optics. As shown in Fig. 1, the Sun sensor consists of two cadmium selenide detectors separated by a shadow vane. The resistance of the detectors is a function of the magnitude of the light intensity with which they are illuminated. Since matched detectors are used, the resistances of the two detectors are equal when the shadow vane points toward the Sun. Two Sun sensors are required on the spacecraft; one for "left-right" motions and the other for "up-down" motions as one looks toward the Sun. A photograph of the Sun sensors and the Sun finders is shown in Fig. 2. The Sun finders actually consist of more light detectors of the same type as are used in the Sun sensors. In fact, the Sun finders are connected as a part of the same circuit with the Sun sensors, the only difference being that the Sun-finder detectors are positioned around the surface of the vehicle so as to provide a spherical field of view for the entire group of detectors. A distinction is made between the two classes of devices (i.e., the "Sun sensor" and the "Sun finder") because, in more advanced spacecraft, there will probably be a simple Sun finder, such as is employed in Ranger, plus an accurate Sun sensor which uses a telescope and other techniques required to achieve an accuracy on the order of 5 sec of arc. Figure 3 indicates the way in which the detector is shadowed when the sensor is pointed directly at the Sun.

Electrically, the detectors are connected in a bridge circuit, as shown in Fig. 4. Cadmium selenide detectors were selected because of their low resistance. Using these detectors, an output of 8 volts per deg is obtained across a 50,000-ohm load without amplification. The Sun sensor has a null stability of about 0.02 deg from day to day. The largest null shift observed to date (due to temperature) has been 0.06 deg.

B. Earth Sensors

The function of the Earth sensor is similar to that of the Sun sensor, except that it must detect and point at the Earth, rather than at the Sun. The Earth does not radiate sufficient energy to allow the use of as simple a device as is employed in the Sun sensor. For lunar missions, the shortcomings of the Sun sensor can be corrected for this Earth-sensor application by one of two methods: the cadmium selenide detectors can be replaced with more sensitive detectors, such as photomultiplier tubes; or optics may be used to concentrate the energy collected over a larger area onto a detector. For the Ranger system, the former approach has been used, and the Earth sensor which has been developed for this application is shown in

Fig. 5. It is interesting to note that both optics and photomultiplier tubes (or equivalently sensitive detectors) would be required for a star sensor.

The Earth sensor consists of three 3/4-in.-diameter photomultiplier tubes, with their field of view constrained by a cover mask in which slits are cut, as shown in Fig. 5. The Earth sensor is a two-axis device which yields information about angular deviations around two axes. It seeks the center of reflected sunlight in the case of a partially illuminated Earth. The manner in which the shadowing technique functions in the Earth sensor can be seen in Fig. 5. If the output signals from the three photomultiplier tubes are designated as A, B, and C, then A - B will give the required error signal for one axis and A + B - C will give the required error signal for the second axis. The Earth sensor has a 30- by 20-deg field of view. As the Earth moves away from the null position, there is a 2.5-deg linear range in either direction; after these points, the system saturates and holds its output for the remainder of its field of view. The sum signal, A + B + C, is used in a closed control loop to adjust the excitation voltage for the photomultiplier tubes. This loop controls the output of the tubes to a constant scale factor (mv/deg off null) independent of range. A photograph of the Earth sensor is shown in Fig. 6. The null stability of the Earth sensor is about ± 0.2 deg.

C. Gyroscopes

The *Ranger* spacecraft employs three gyroscopes to orient the vehicle in any arbitrary direction with respect to the Sun and Earth reference directions for short periods of time (approximately 1 hr). The gyros are connected as rate gyroscopes, and, in order to orient the entire spacecraft away from its zero reference position, a constant rate of turn may be commanded for a controlled period of time about each of three mutually orthogonal directions. The new position is maintained by using the three gyros to detect any angular rates which may occur after the turns are completed and nulling these rates to zero with the attitude-control actuators. In Ranger, two requirements make it necessary to turn the entire spacecraft away from its zero reference position: (1) to fire the midcourse rocket motor which is rigidly attached to the vehicle; (2) to eject a capsule in the vicinity of the Moon (one of the scientific experiments is to land a capsule on the surface of the Moon).

The package of three gyros used on the Ranger spacecraft is shown in Fig. 7. These are ordinary single-degree-of-freedom floated gyroscopes which have been slightly modified for this space application. In order to conserve electrical power in the spacecraft,the floatation fluid has been changed to a low-viscosity fluorolube requiring no heaters. Also, the spin motor bearing preload has been reduced to provide all possible increase in life expectancy while still retaining the rigidity necessary to survive the launch environment.

For those who may not be familiar with the single-degree-of-freedom floated gyroscope, it is briefly described herein. As indicated in Fig. 8a, the spinning rotor of the gyro is contained within a hollow, closed structure which, in turn, is contained within the outer case of the instrument. The space between the rotor structure and the outer case is filled with a fluid. The density of this fluid is so adjusted that the rotor structure is perfectly floated (or is "neutrally bouyant") in the fluid; hence, the rotor structure is often called a "float." In any gyroscope, the suspension which supports the rotating member must be designed in such a way that it exerts nearly zero torque on the rotor. The purpose of the floatation is to provide a high-quality suspension, which is accomplished by the fact that the pivots at each end of the float must support very little weight, and, in the ideal case of perfect floatation, they act merely as location devices.

An angular rate of rotation about the input axis tends to cause the gyro rotor to precess about the output axis. However, as soon as the float starts to turn about the output axis, this motion is sensed by a position pickoff device. The electrical signal from this pickoff device is amplified, and the output of the amplifier drives a torquing device, as shown in Fig. 8b. The torque produced by this device is proportional to the current flowing through it, and the torque acts about the output axis. The closed loop shown in Fig. 8b is connected in such a way as to have negative feedback and acts to maintain the pickoff device very close to its null position. In order to prevent the pickoff from turning continuously, the torque applied by the torquer must cause the rotor to precess about the input axis at a rate that makes the float follow the rotations of the instrument case about the input axis. In the steady state, the torque produced by the torquer is proportional to the input rate of rotation. Hence, the current flowing through the torquer is also proportional to angular input rate, and a voltage proportional to angular rate is obtained across a series resistor, as shown in Fig. 8b.

During the cruise and acquisition phases, the gyros are connected exactly as described in the previous paragraph in order to provide a voltage proportional to angular rate. However, during periods of commanded turns away from the zero reference position of the Sun and Earth sensors, a different circuit is required. During periods of commanded turns, the Sun and Earth sensors are disconnected, and the gyro torquing device (for each gyro in turn as required) is driven by a constant current source for a controlled period of time, as indicated in Fig. 9. This scheme essentially commands a constant rate of turn, since the gyro loop amplifier must deliver a current (which causes an output across the series resistor) unless the gyro is rotating about its input axis at a rate corresponding to the constant current. If the system were perfect, that is, if the vehicle had no inertia and if there were no disturbing torques, the circuit illustrated in Fig. 8b would be satisfactory. However, the system is not perfect; therefore, a signal proportional to the position error of the vehicle must be provided during the period of commanded turns. This signal is obtained by placing a capacitor in series with the gyro torquer, as shown in Fig. 9. Since the current flowing through the torquer is proportional to angular rate, the voltage across the capacitor (which is proportional to the integral of the current) must be proportional to position. This position signal is zero if the angular position of the vehicle is at its desired value as required by the constant turn rate. Any deviation from the desired position causes a voltage to appear across the capacitor which drives the attitude control actuators in such a way as to correct the position error.

It is interesting to note that, if the gyro floatation fluid had a sufficiently high viscosity, the gyro could be used in an open-loop manner with the integration to position provided by viscous integration with the fluid. As was mentioned previously, a very low viscosity floatation fluid was used in the Ranger gyros in order to avoid the requirement for gyro heaters.

D. Horizon Seekers

The horizon-seeker instrument (not used on Ranger) is applicable where it is desirable to find the direction toward the true center of a planet, rather than toward the center of radiation or reflected light. As its name implies, a horizon seeker detects the boundary between the edge of a planet (or some part of the atmosphere if one exists) and outer space. There are many different types of horizon seekers, but perhaps they can all be classified as similar to one of the three types to be described below.

One type of horizon seeker merely detects the horizon at three fixed points and makes use of the fact that the locations of three points on the circumference of a circle are sufficient to determine the location of the center of the circle. Although it is most convenient to use three points separated by 120 deg, this technique can also be used (with the aid of a computer) to determine the center when only the locations of three points on the outer edge of a crescent are available.

A second type of horizon seeker uses a scanning operation (either pre-image or post-image variety) to determine continuously the location of the entire edge of the disc. This type of device often utilizes detectors which are sensitive in the far infrared region. In this spectral region, the planet always appears as a complete disc (although not of uniform intensity because of temperature variations over the surface of the planet) as a result of the black-body or temperature radiation from the planet. The spectral region containing reflected sunlight is usually filtered out.

A third type of horizon seeker uses an occulting technique in which all but the edge of the image is blanked out by an opaque barrier or stop in the instrument. This technique is most easily applied to determining the direction toward the center of the Sun.

E. Star Sensors

Although not used on Ranger, a star sensor may be required for attitude-control purposes on more advanced planetary missions. A star is effectively a point source; therefore, its direction is considerably easier to determine than is the direction toward the center of an extended body such as the Sun or a planet. The relatively low intensity of a distant star is not a problem if a photomultiplier tube is used as the detector. However, there is a problem in distinguishing a single star, which may be desired as an attitude reference, from all other stars in the sky. A star is most simply identified by its brightness. Often, brightness alone will be sufficient since considerable advantage may be taken of the known geometry of the situation in order to eliminate all other stars of brightness similar to that of the desired star. In the event that absolute brightness alone is not adequate, relative brightness in two spectral "pass bands," such as the red and the blue, can be used as an additional criterion of identification. More elaborate identification schemes have been proposed, one of which uses absorption spectra as a means of identification and another, pattern recognition of a particular group of stars.

For stellar identification, schemes have often been proposed which would allow a great number of directions, providing coverage over an entire sphere, to be recognized. Such schemes involve storage of "star maps" to be compared with observed patterns, or of "star catalogues" to be compared with observed spectral characteristics. It should be pointed out that this ability to recognize many directions around an entire sphere is an unnecessary complication for missions within the solar system. A much simpler method is always to acquire the Sun first, after which the identification of one other direction yields a complete coordinate reference system.

III. ACTUATORS

Some actuation techniques and their capabilities, which are applicable to space-vehicle attitude control,are listed in Table 3. Only the cold-gas jet system, used on the *Ranger* spacecraft, will be considered in detail.

A. Cold-Gas Actuation System

A diagram of the *Ranger* actuation system is shown in Fig. 10; it is a cold-gas system which is pressurized on the ground prior to flight. The components comprising the system are the gas storage tank, regulator, and ten nozzle-valves with interconnecting piping. Each nozzle-valve is an integral assembly consisting of a nozzle plus an electrically operated valve which is opened when it is desired to rotate the spacecraft. The nozzle is a separable part in order that it may be changed as required to provide different thrust levels. A valve having a metallic seat was selected because of the unknown effects of the space environment on plastics. The nominal thrust of each nozzle is about 0.005 lb.

The gas which is used in the *Ranger* actuation system is dry nitrogen, which was selected because of its relatively high specific impulse, its chemical inertness, and its ability to act as a lubricant between moving parts. At its operating temperature of 80°F, the nitrogen will have a specific impulse of 76 sec (lb of force/lb/sec of gas consumption). A total of 2.5 lb of nitrogen is stored at a pressure of 3000 psi for the *Ranger* system, which is intended for use in a lunar mission requiring 66 hr of operation. The storage tank is a spherical pressure vessel approximately 8 in. in diameter. The pressure regulator reduces the pressure of the nitrogen gas to 15 psi before it is admitted to the valves. The gas leakage for the entire system will be less than 0.05 lb/day.

In the Ranger system, the valves are arranged in pairs and provide a torque level of about 0.02 ft-lb and a vehicle angular acceleration of about 0.01 deg/sec^2 (0.2 mr/sec^2). It is an "on-off" system which has only this one torque level, or zero torque. The gas flow rate for a single pair of valves is about 0.01 lb/min. Thus, the 2.5 lb of gas carried in the storage tank would be sufficient to provide 1.4 hr of continuous operation for each of the three axes. For an actual 66-hr lunar flight, assuming reasonable values for initial separation rates, disturbances, and limit cycle operation during the cruise phases, the quantity of 2.5 lb is about 5 times larger than the predicted gas consumption. As indicated in Fig. 10, the expected rate of gas consumption during limit cycle operation without disturbing torques is about 0.002 lb/hr.

B. Momentum Interchange Devices

Momentum interchange devices employ the principle of action and reaction without expelling any mass from the vehicle. The basic idea can be described by considering a flywheel mounted in the vehicle. If the flywheel is given an angular acceleration in one direction, an angular acceleration is produced on the entire vehicle in the opposite direction. Since the action and reaction torques are equal, and opposite in direction, the spacecraft is accelerated at a rate dependent on the ratio of its moment of inertia to that of the flywheel. Looking at this from the point of view of conservation of angular momentum, if the spacecraft has an initial angular velocity when it is separated from the injection vehicle, this velocity can be reduced to zero by transferring the momentum to a flywheel which rotates with respect to a fixed spacecraft. Three flywheels, or reaction wheels as they have been called, can be used to control the attitude of a vehicle about three axes. When this is done, gyroscopic precession torques must also be considered, and probably used as a part of the actuation scheme. A sphere, suspended either electrostatically or by an air bearing, and torqued about three orthogonal axes, can also be used for the momentum interchange type of actuation. In this mechanization, no gyroscopic precession torques act on the vehicle.

Two problems associated with momentum interchange devices are: (1) the motor (and the flywheel which it drives) will always have a maximum angular velocity that cannot be exceeded, thus the capability of the actuators is always limited; (2) friction losses which consume power are undesirable. The friction problem can be eliminated by using nearly frictionless bearings. However, the only practical solution to the velocity saturation problem is to supplement the momentum interchange device with some other type of actuator which can be used to periodically reset the momentum wheel or sphere back to zero velocity. A cold-gas system as used in Ranger could be used for this purpose; however, a hot-gas system has characteristics which complement those of momentum interchange devices. A hot-gas system utilizing solid fuel eliminates the gas-leakage problem associated with cold-gas actuators. However, because of corrosion problems, hot-gas systems cannot be operated for long periods of time. The application of hot-gas techniques to reset infrequently the momentum interchange devices appears to yield a desirable system.

C. Solar Pressure Devices

The solar-radiation pressure in space at the distance of the Earth from the Sun is about 5×10^{-5} dynes/cm.^{2} Although this pressure is too low to be used as the only means of actuation, solar pressure could be employed by utilizing movable "vanes" or "sails" to obtain torques in the directions necessary to reset momentum interchange devices. In the use of very low torque levels to reset momentum interchange devices, it is interesting to note that a very special type of "cold-gas" system could be employed; namely, the vapor pressure of an evaporating liquid or of a sublimating solid.

The Ranger actuation system was selected on the basis of weight (it provided a system having the lowest weight) and partly because it required the shortest development time. As the time of flight is increased (for planetary missions, rather than lunar missions), momentum interchange devices become more desirable from the aspect of weight restrictions.

IV. THE *RANGER* ATTITUDE-CONTROL SYSTEM

A photograph of a model of the *Ranger* spacecraft is shown in Fig. 11. The mission of the *Ranger* spacecraft is to perform certain scientific experiments in the vicinity of the Moon. However, only those aspects of the vehicle associated with its attitude control during the time prior to its arrival in the vicinity of the Moon are considered here. The attitude-control requirements in the vicinity of the Moon may be very specialized, depending upon the nature of the scientific experiments which are performed. In order to present material of general applicability, only three of the attitude-control requirements will be considered:

a. Orienting the solar cells toward the Sun,

b. Directing the communications antenna toward Earth,

c. Pointing the thrust vector of the midcourse rocket in the desired direction.

A discussion of these attitude-control requirements involves two active modes: (1) the initial acquisition period when the spacecraft is released from the launching vehicle and first acquires the Sun and Earth; (2) the midcourse-maneuver period when a rocket is fired to correct the trajectory of the vehicle. Each of these periods occupies a time of about 1 hr or less, and they are separated by a cruise period of about 15 hr.

The midcourse maneuver is followed by a second cruise period of about 50 hr, until the vehicle arrives in the vicinity of the Moon. A flight sequence indicating the items of interest for attitude control is shown in Table 4.

A. The Acquisition Phase

Electrical power for the Ranger spacecraft is supplied by solar cells for most of the flight and by a battery during those periods of time when the solar cells are not oriented toward the Sun. The solar cells are mounted on two hinged panels (henceforth called the "solar panels"), which are extended after the spacecraft separates from the injection vehicle. The antenna is mounted on an arm with a similar single-degree-of-freedom hinge, and it also is unfolded or extended after the separation of the spacecraft from the injection vehicle. The locations of the rocket motor, the solar panels in various positions, and the antenna in various positions, are shown in Fig. 11, 12, and 13. These Figures also show the location of the Sun sensors, the Earth sensor, and the cold-gas nozzles.

When the spacecraft separates from the injection vehicle, it may be spinning about an arbitrary direction at a rate between 1000 and 10, 000 deg/hr. The first function of the attitude-control system is the reduction of this initial spin rate to zero, which is accomplished by sensing the rates about three orthogonal axes with three gyroscopes and using the gyro signals to cause the actuators to apply a torque which reduces the angular velocity to zero. The jet nozzles operate continuously during this period, and 0. 5 lb of the available 2. 5 lb of nitrogen gas would be consumed if the initial spin rate were 10, 000 deg/hr.

When the angular rate has been reduced, the signal from the Sun finder/Sun sensor circuitry takes over. This causes the actuators to apply torques which orient the Sun sensor towards the Sun. When both Sun sensors have locked onto the Sun, the direction of a line in space, that is, the line from the spacecraft to the Sun, is established. The roll axis of the spacecraft lies along this line when the error signals from both Sun sensors are zero. The pitch axis of the spacecraft is parallel to the antenna hinge axis, and the yaw axis is parallel to the solar-panel hinge axes. After the Sun has been acquired, the vehicle is stabilized in pitch and yaw. It then rolls about the line from the spacecraft to the Sun until the Earth sensor "sees" the Earth and locks onto it. The Earth sensor is fixed to the vehicle with respect to motions about the vehicle roll axis; thus, the angular position of the vehicle about the roll axis is determined when the Earth sensor locks onto the Earth. However, since the Earth sensor is mounted on the same movable arm which supports the antenna, it is not fixed with respect to motions about the vehicle pitch axis. This is done for two reasons: (1) the arm supporting the antenna and Earth sensor is erected at a preset angle with respect to the vehicle roll axis during the initial acquisition phase (based on a prior knowledge of the geometry of the trajectory) in order that the Earth will fall within the field of view of the Earth sensor when the spacecraft turns about its roll axis after acquiring the Sun; and (2) the "pitch" error signal from the Earth sensor is used in a servo loop to drive a motor and gear train which maintains the antenna pointed at the Earth by providing the proper motions about the antenna hinge axis (pitch axis of vehicle).

B. The Cruise Phases

After the Sun and Earth have been acquired, the acquisition phase is completed, and the first cruise phase is started. During this phase, the attitude-control loop involving the Earth and Sun sensors and the actuators operate in a limit cycle mode in order to conserve the nitrogen gas supply.

A simplified block diagram, typical of the attitude-control servo loop for any of the three axes, is shown in Fig. 14. As mentioned previously, three rate gyros are provided to measure angular rates about the three major axes of the spacecraft. The angular position signal is provided by the Sun sensors for the pitch and yaw axes, and by the Earth sensor for the roll axis. The switching amplifier is a transistor amplifier designed to have a deadband and hysteresis as indicated in Fig. 14. The deadband characteristic causes the valve and nozzle to be "full on" whenever the input (the sum of the rate gyro plus the position sensor signal) to the switching amplifier is greater than a certain value preset within the switching amplifier. This type of operation causes the attitude-control system to operate in a stable limit cycle during the cruise phase and is desirable for the purpose of conserving gas. During the initial acquisition phase, when the spacecraft is tumbling, the rate gyro output signal overpowers the position signal and causes the appropriate valves to remain open continuously. The rate channel gain increases whenever the gyro signal exceeds a certain threshold level determined by zener diodes. Thus, a continuous torque acts to retard the rotation of the vehicle and to reduce its spin rate to zero. However, during the cruise phase, when the rates are very low and the vehicle is very close to the desired attitude, the valves are closed and no gas is used. When the angular position of the spacecraft drifts away from the desired attitude by an amount such that the position signal exceeds the threshold of the switching amplifier, the valve is opened momentarily and a pulse of torque is applied to the spacecraft. The duration of the pulse of torque is determined by the gain of the rate gyro and the amount of hysteresis in the switching amplifier. The hysteresis is employed to prevent valve "chatter," and thus increase the reliability of the valve and conserve gas. In the Ranger system, the dead zone during the cruise period corresponds to $\pm$ 1 mr ($\approx$ 0.05 deg) of position, or $\pm$ 2 mr/sec ($\approx$ 0.1 deg/sec), of angular velocity, or any appropriate combination thereof.

To summarize, during the cruise phase the spacecraft remains within the deadband most of the time. As the spacecraft leaves one side of the deadband, the amplifier turns on instantaneously. The amplifier output opens the valve, and torque is applied to the spacecraft for only a short period of time required for the rate gyro signal to drive the amplifier input through its hysteresis zone. When this occurs, the amplifier then turns off instantaneously. A typical torque pulse has a duration of 0.1 sec.

During the cruise phase, it is certainly not required that the attitude of the vehicle be controlled to within $\pm$ 1 mr (the deadband described above). This order of accuracy is required only immediately prior to the start of the commanded turns in order that the gyro may produce measured turns from an accurate reference position. The tight deadband of $\pm$ 1 mr is used throughout the flight (rather than switching to a wide deadband during the cruise phase) because the gas consumption may be determined by a factor other than the switching amplifier deadband; that is, solar pressure and other disturbances may produce torques on the spacecraft. Thus, the repetition rate of the applied pulses of torque (or, more exactly, the integrated torque on the vehicle due to the gas jets) must be such as to balance these disturbance torques in the steady state. If the deadband of the switching amplifier is made very wide, a mode of operation in which the vehicle remains near one side of the deadband will occur and no gas will be saved by the wide deadband. For this reason, and since there is plenty of gas available, rather than changing the deadband, it is set at $\pm$ 1 mr for the entire flight. Three other known sources of attitude disturbances (in addition to solar pressure) include moving parts in the vehicle, collisions with micrometeorites, and gravitational forces.

The block diagram of the Ranger attitude-control system presented in Fig. 15 shows all three axes of attitude control, the antenna control, and the constant current source for the commanded turns.

The angular rate signals required to stabilize the limit cycle operation can either be obtained directly from gyros, as described above, or derived from the valve operating voltages. However, since the gyros are required to remove the initial spin rate, they are also used for the limit cycle operation. For planetary missions with longer lifetime requirements, if a cold-gas type of system were employed it would probably use derived rate signals during the long cruise phase in order that the gyros would not have to operate continuously. The gyros would be turned on only during active periods.

C. The Midcourse Maneuver

After cruising for about fifteen hours (at which time it is at a distance of about 10^5 miles from the Earth), the spacecraft is ready to execute a midcourse-maneuver command which has been transmitted via the radio link and stored in a memory device contained within the spacecraft. This stored command contains four items of information. It contains the magnitudes and directions of two angles through which the spacecraft must turn (one angular turn about the roll axis and another about the pitch axes) in order to point the thrust vector of the rocket motor (which is fixed to the spacecraft) in the desired direction. The other two items of information are not directly concerned with attitude control, but include the time at which the rocket motor should be started and the velocity increment which must be imparted to the spacecraft. The velocity increment is controlled by means of an accelerometer and integrator which generate a shut-off signal for the rocket.

As indicated previously, the gyros are connected as rate gyros rather than position gyros, and the turns are accomplished by commanding a constant rate of turn at 0.2 deg per sec for a specified period of time. The Earth and Sun sensors are disconnected during the commanded turns, and position signal inputs for the switching amplifiers are obtained by including a capacitor in series with each gyro torquer coil.

While the rocket motor is being fired, the cold-gas nozzles do not have sufficient torque to control the vehicle in the event that the rocket thrust exerts a torque on the vehicle. To provide attitude control, or more precisely, thrust vector control (since what is actually desired is to have the thrust vector of the rocket simultaneously point in the desired direction and pass through the center-of-gravity of the vehicle, regardless of the exact orientation of the vehicle itself) during this period, jet vanes are employed in the exhaust path of the rocket motor. Separate autopilot electronics with error signal inputs derived from the three gyroscopes and the jet vane positions are also used during this period.

After the midcourse maneuver is completed, the Sun and Earth are re-acquired by repeating the same process associated with the initial acquisition. The spacecraft then enters a second cruise period, and goes on to accomplish scientific experiments in the vicinity of the Moon.

Table 1. Requirements for space-vehicle attitude control (rms errors)

Function	Accuracy, deg
Orient solar cells toward Sun	10
Temperature control	10
Direct antenna toward Earth	2
Scientific experiments	0.5
Midcourse maneuver	0.25

Table 2. Sensors for space-vehicle attitude control (rms errors)

Item	Application	Accuracy, deg		
		Usable	Desired[a]	Ranger
Sun finder	locate direction toward Sun from any initial orientation	— not applicable —		
Sun sensor	indicate angular deviation from direction toward center of Sun	1	0.001	0.1
Planet sensor	indicate angular deviation from direction toward center of radiation of a planet	1	0.03	0.2
Horizon seeker	indicate angular deviation from direction toward true center of a planet	1	0.001	not applicable
Star sensor	indicate angular deviation from direction toward a star	1	0.001	not applicable
Gyroscope	(1) short-term reference for use during commanded turns of the spacecraft away from celestial references (2) to measure angular rates of turn	3/hr	0.001/hr	0.2/hr

[a]Desired high accuracies reflect guidance requirements when the same sensor is used for both guidance and control.

Table 3. Actuation techniques for space-vehicle attitude control

Technique	Capability	Force Characteristic	Useful Torque Range, ft - lb	Application	Comments
Mass expulsion	cold gas	I_{sp} = 80 sec	0.001 to 25	complete system	simple; well developed; leakage a problem for long missions; storage tank is heavy.
	hot gas	I_{sp} = 275 sec	0.01 to 80	reset momentum device or complete system	presently useful only for intermittent short-term operation because of corrosion.
	vapor pressure	I_{sp} = 80 sec	0.001 to 0.1	reset momentum device or complete system	new; requires large solar-heating area or electrical power; storage tank is lightweight.
Momentum interchange	reaction wheels	—	0 to 1	used with auxiliary reset mechanism	new; provides continuous control (no limit cycle).
	reaction sphere	—	0 to 0.3	used with auxiliary reset mechanism	new; provides continuous control (no limit cycle); no precession torques.
Solar pressure (near the Earth's orbit)		10^{-7} lb/ft^2	to 2×10^{-5}	reset momentum device	new; solar sails may interfere with other field-of-view requirements.

Table 4. _Ranger_ flight sequence for attitude-control functions

Item	Time		Function
	hr	min	
1		0	lift-off from launching pad
2		22	spacecraft separated from launching vehicle
3		30	unfold solar panels (explosive charge)
4		33	turn on attitude-control-system power a. extends antenna to preset position b. activates gas jet system c. activates Sun sensors d. commence Sun acquisition
5	3.5		activate Earth sensor start roll search
6	3.5 to 4		Earth sensor locks on Earth a. stops roll search b. starts antenna hinge servo
7	3.5 to 4		roll and hinge over-ride if necessary (repeat 5 and 6)
8	4		acquisition complete
9	4 to 16		cruise period
10	16		start midcourse maneuver sequence a. turn on midcourse autopilot b. turn off Earth sensor c. execute commanded roll turn

Table 4. (Cont'd)

Item	Time		Function
	hr	min	
11	16	9	turn off Sun sensors
			execute commanded pitch turn
12	16	26	start midcourse motor
13	16	27	shut off midcourse motor with accelerometer
14	16	28	turn off midcourse autopilot
			commence Sun acquisition
15	16	88	steps 5, 6, 7, and 8 completed a second time
16	17.5 to 65		cruise period
17	65 to 65.5		perform scientific experiments in vicinity of Moon

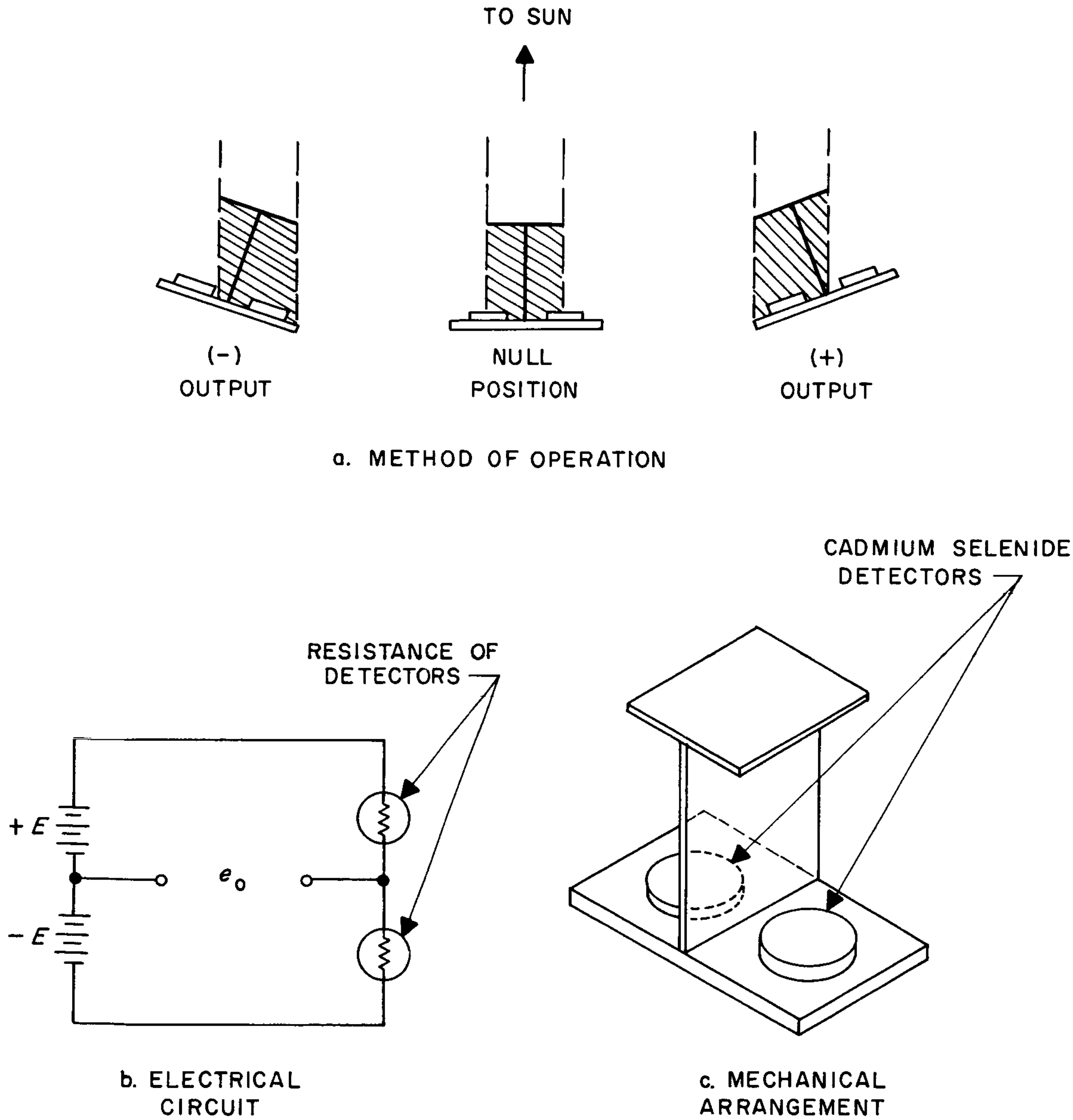

Fig. 1. Sun sensor

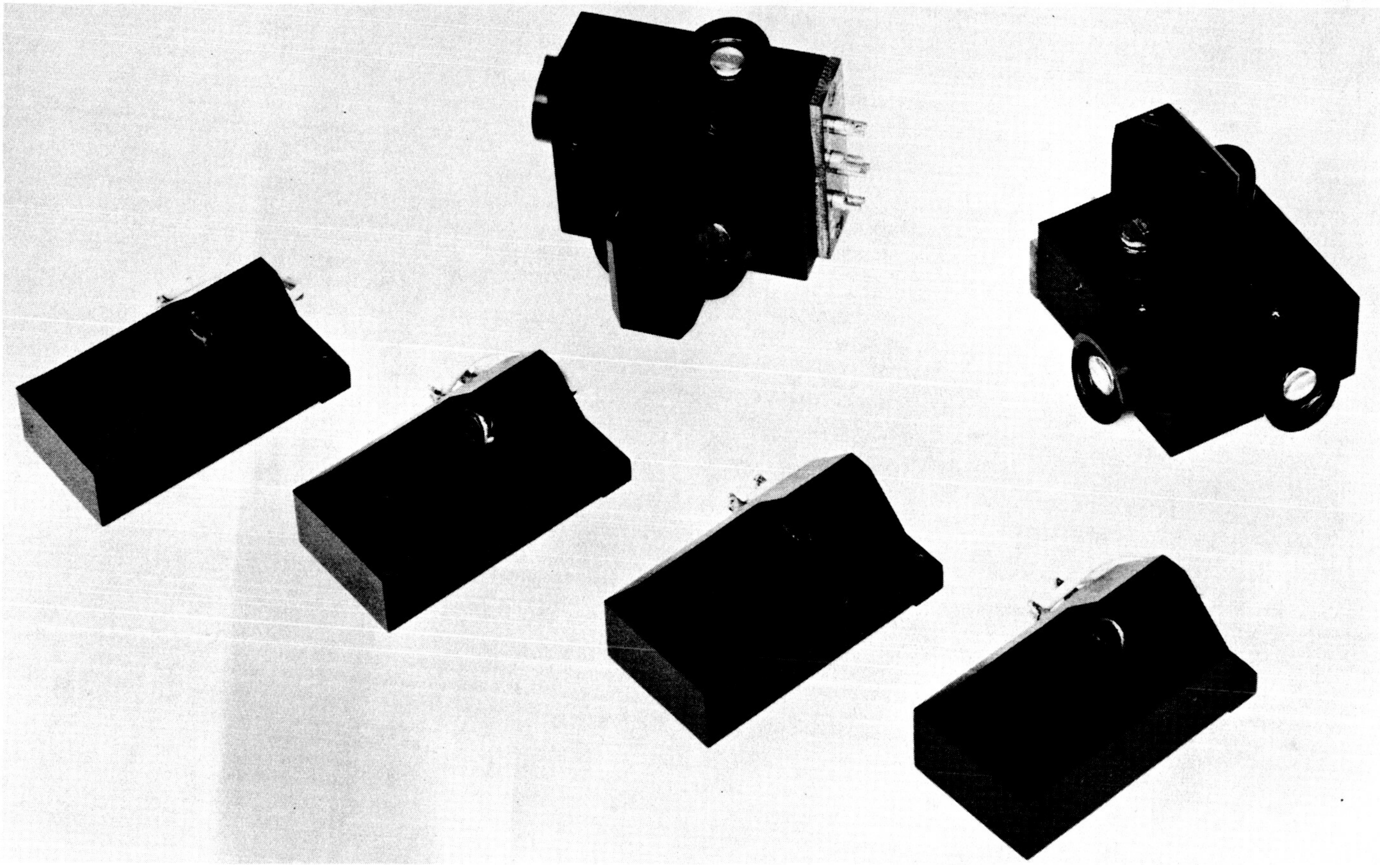

Fig. 2. Sun sensors and Sun finders

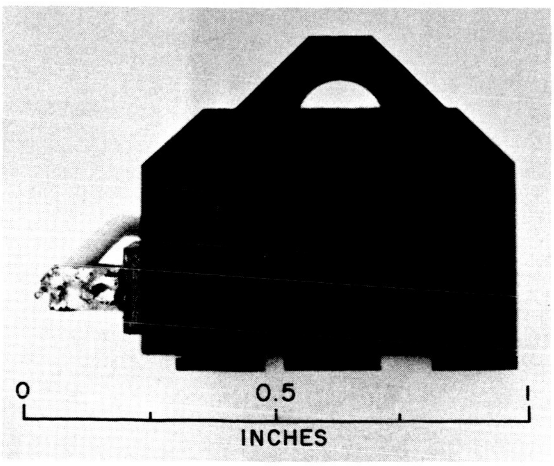

Fig. 3. Sun sensor shadowing technique

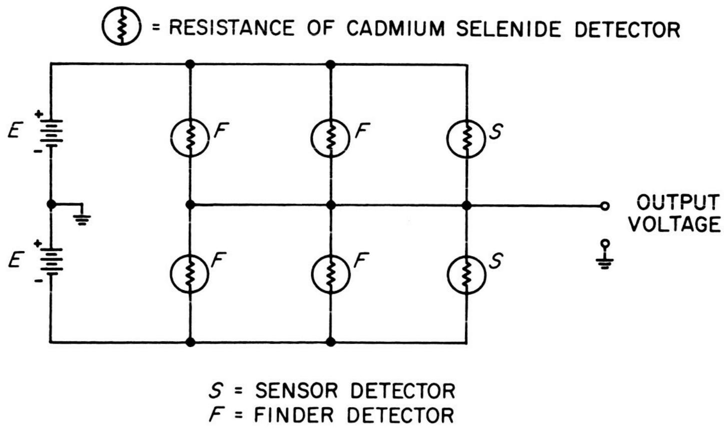

Fig. 4. Sun sensor electrical circuits

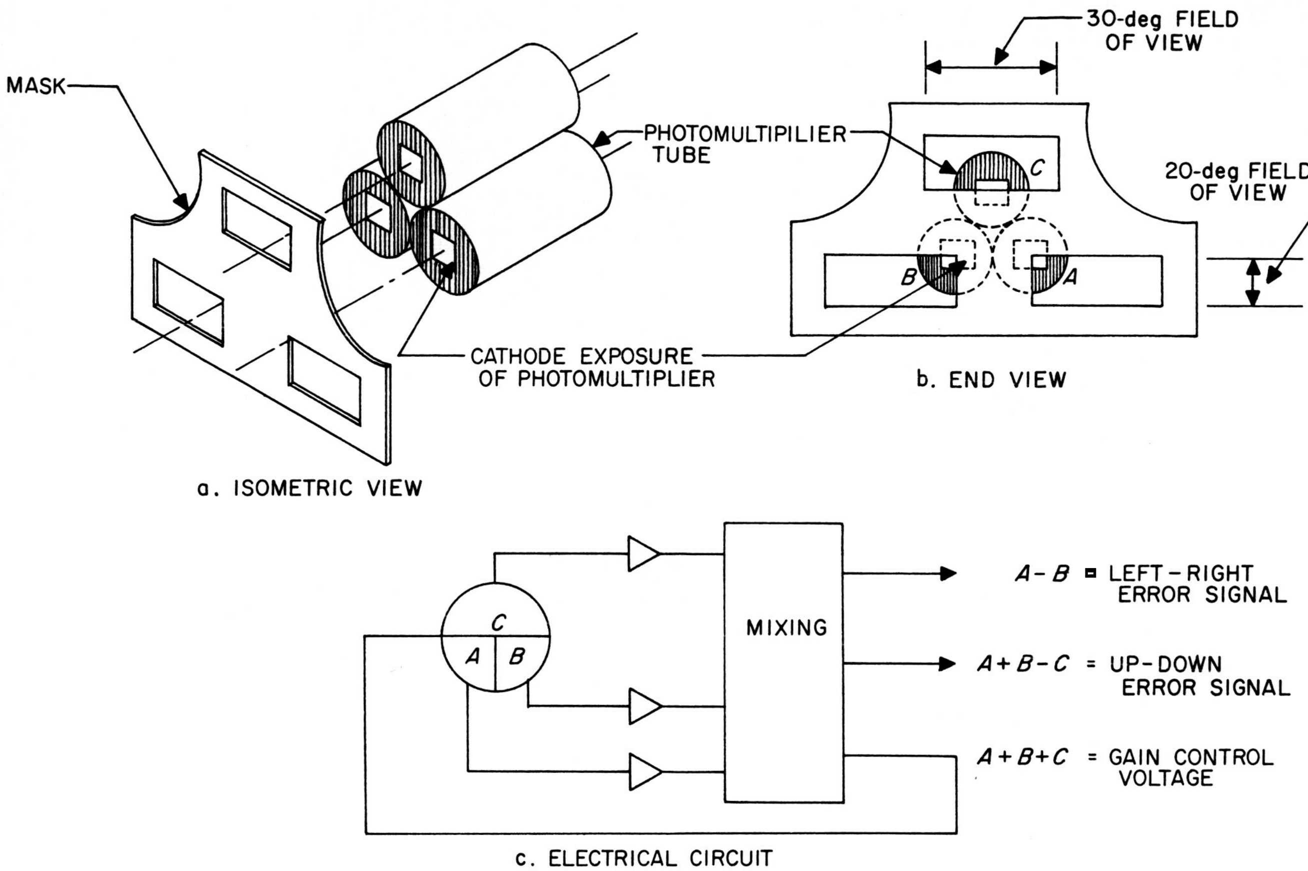

Fig. 5. Earth sensor

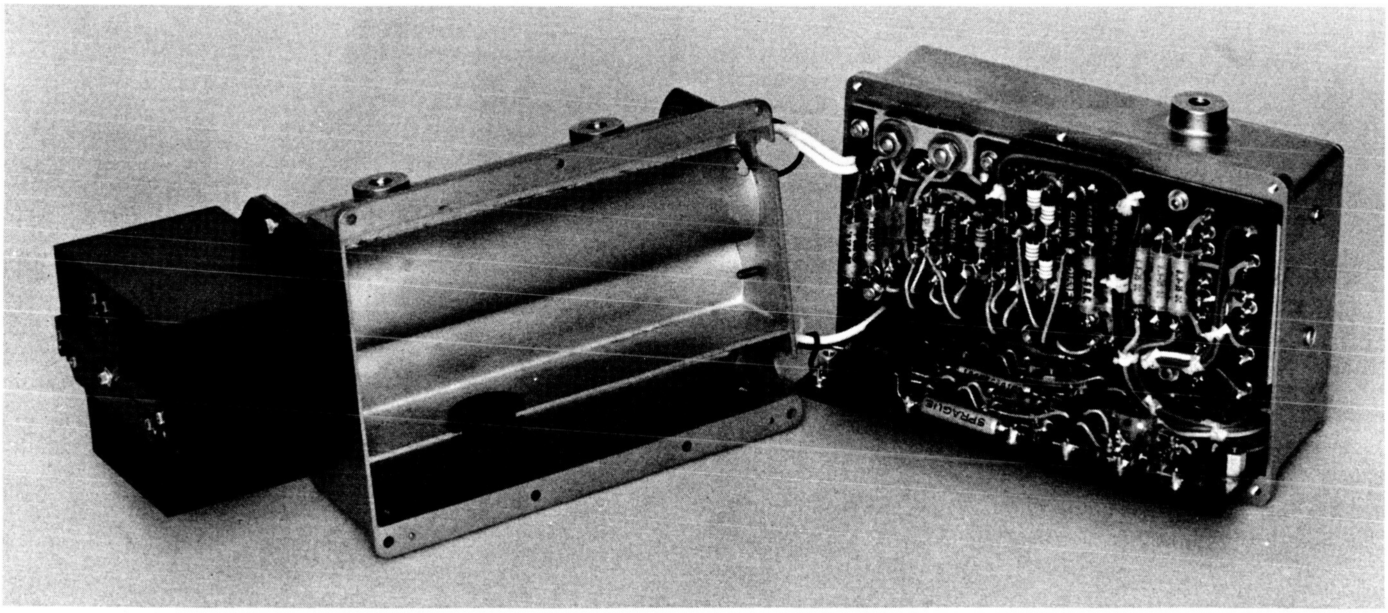

Fig. 6. Photograph of Earth sensor

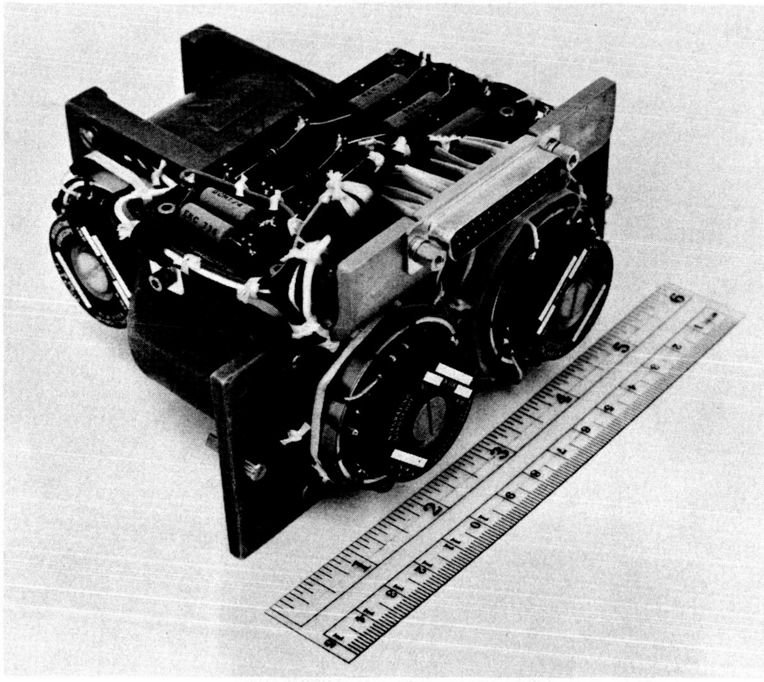

Fig. 7. Ranger gyro package

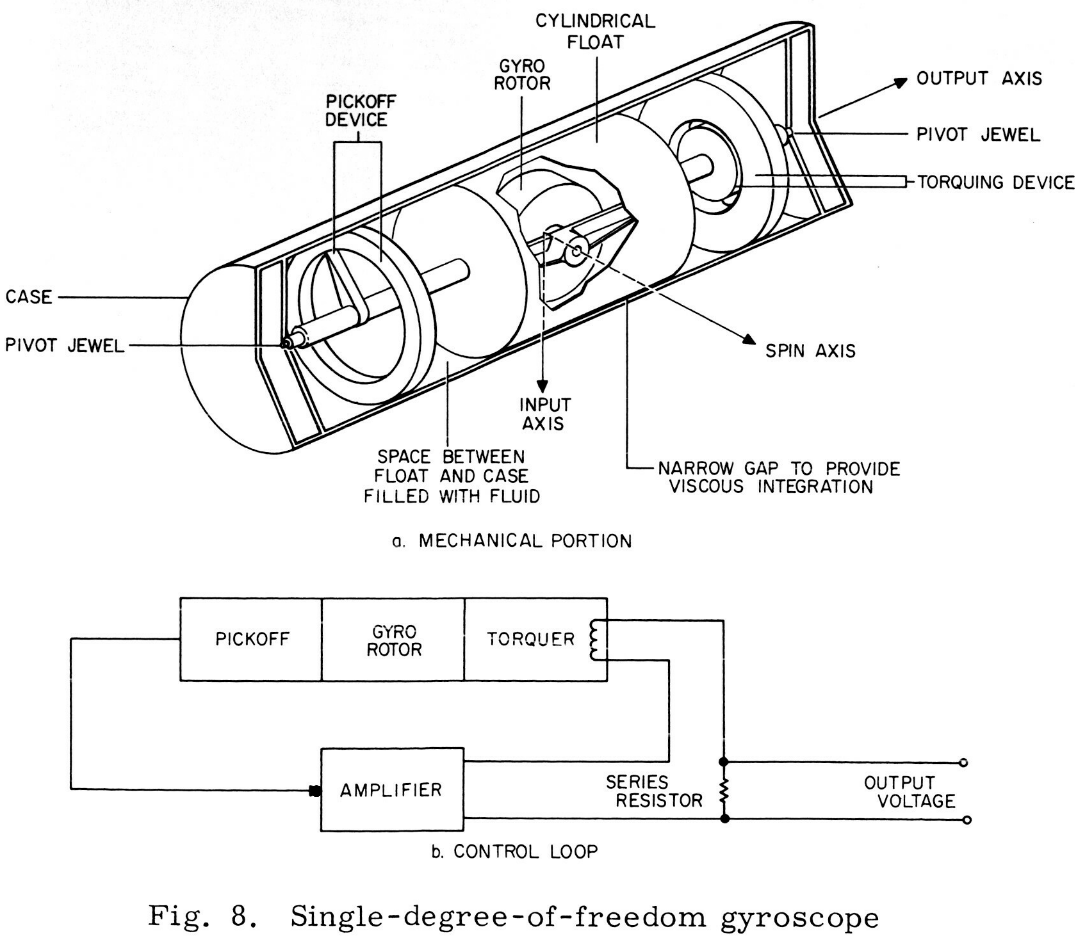

Fig. 8. Single-degree-of-freedom gyroscope

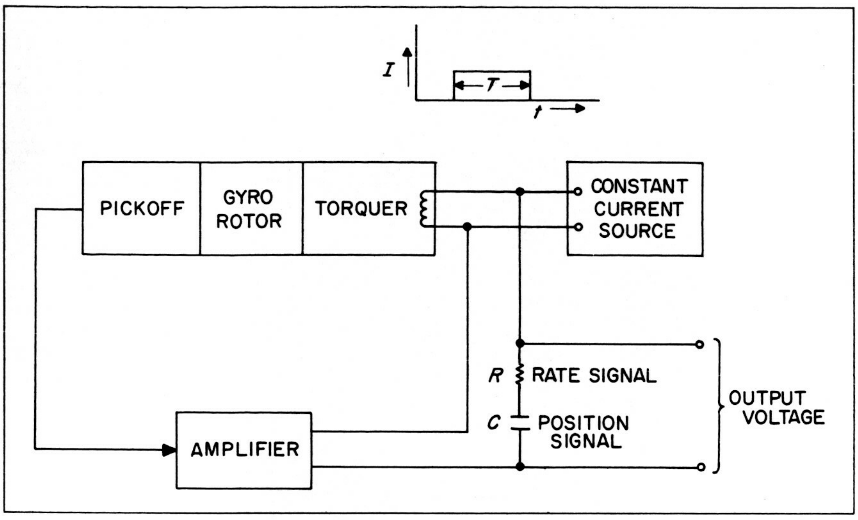

Fig. 9. Gyro control loop during commanded turns

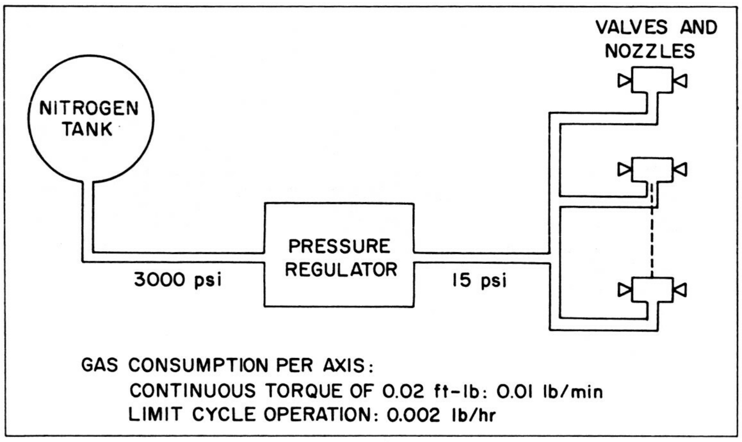

Fig. 10. Cold-gas actuation system

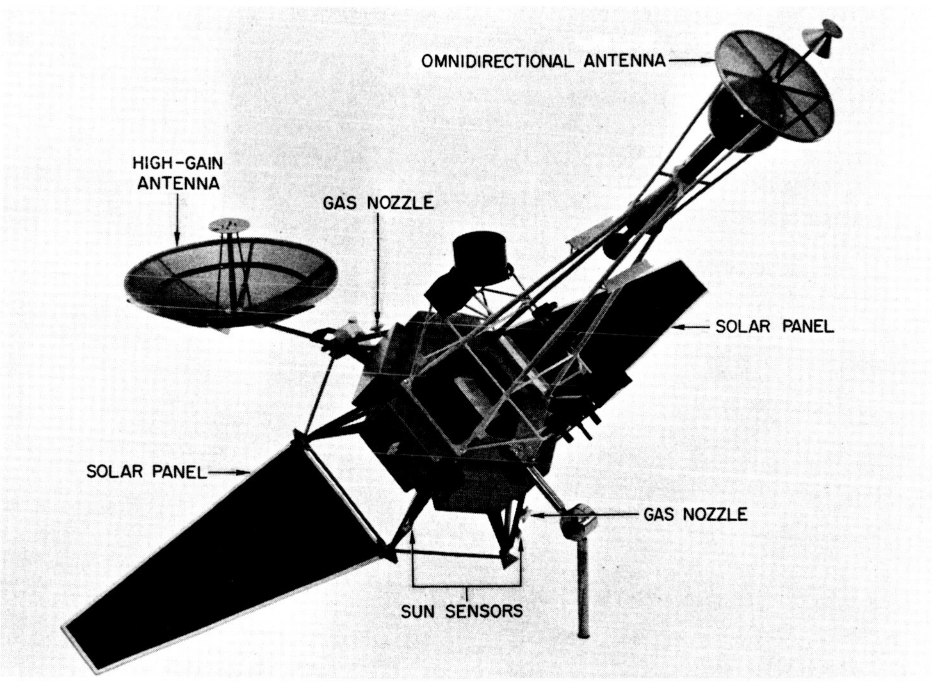

Fig. 11. Ranger spacecraft, right-front view

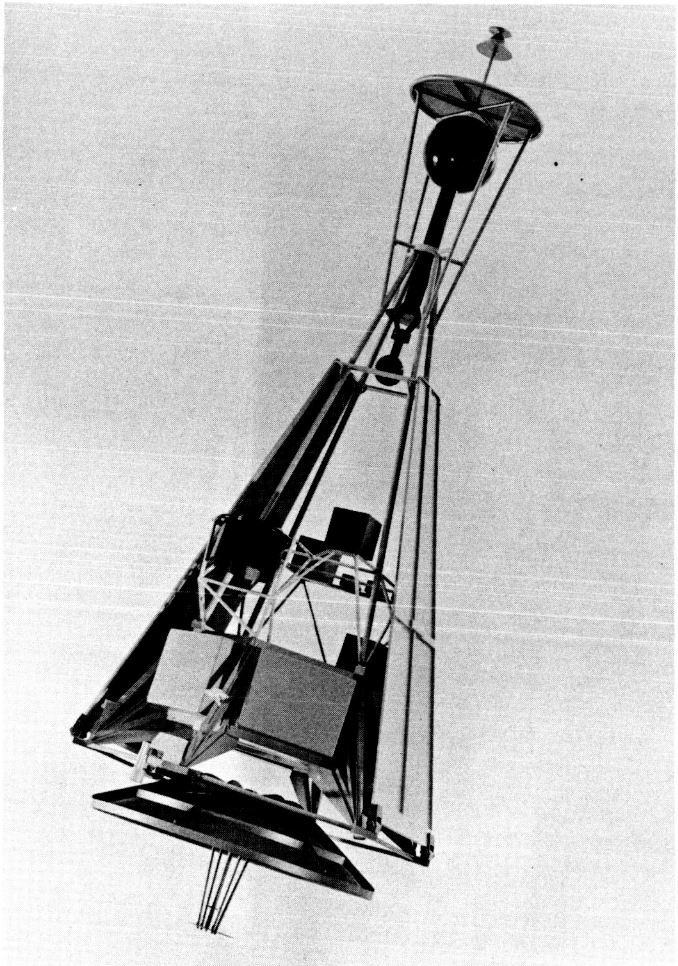

Fig. 12. Ranger spacecraft, solar panels folded closed

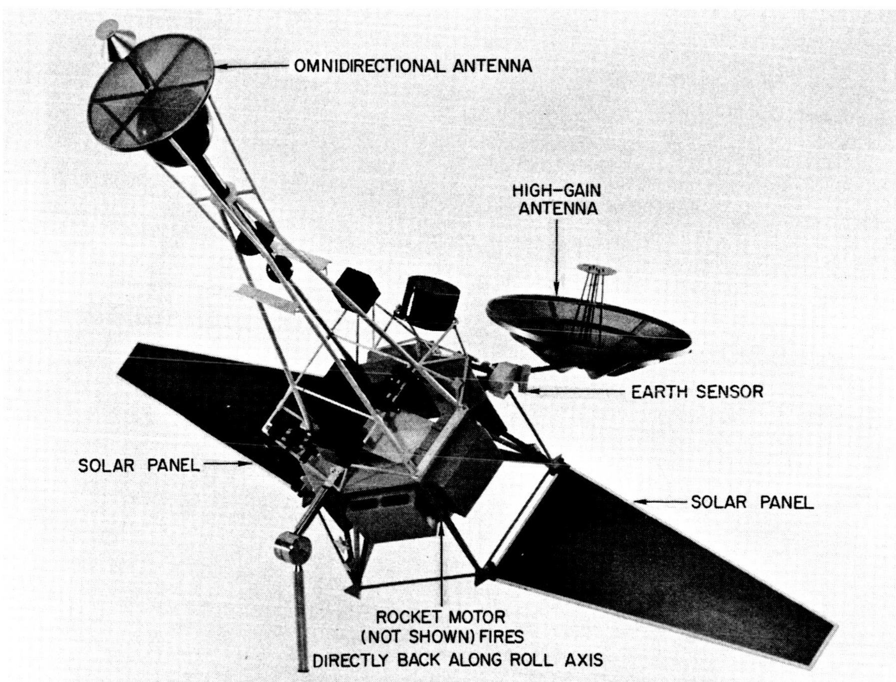

Fig. 13. Ranger spacecraft, left-front view

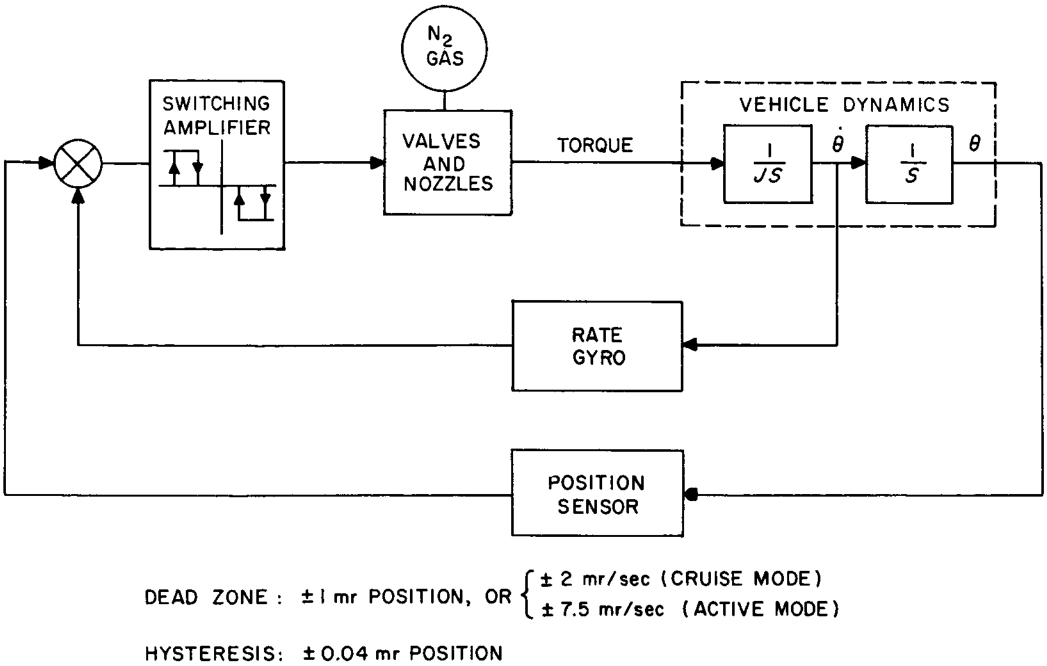

Fig. 14. Ranger attitude-control loop (typical of any axis)

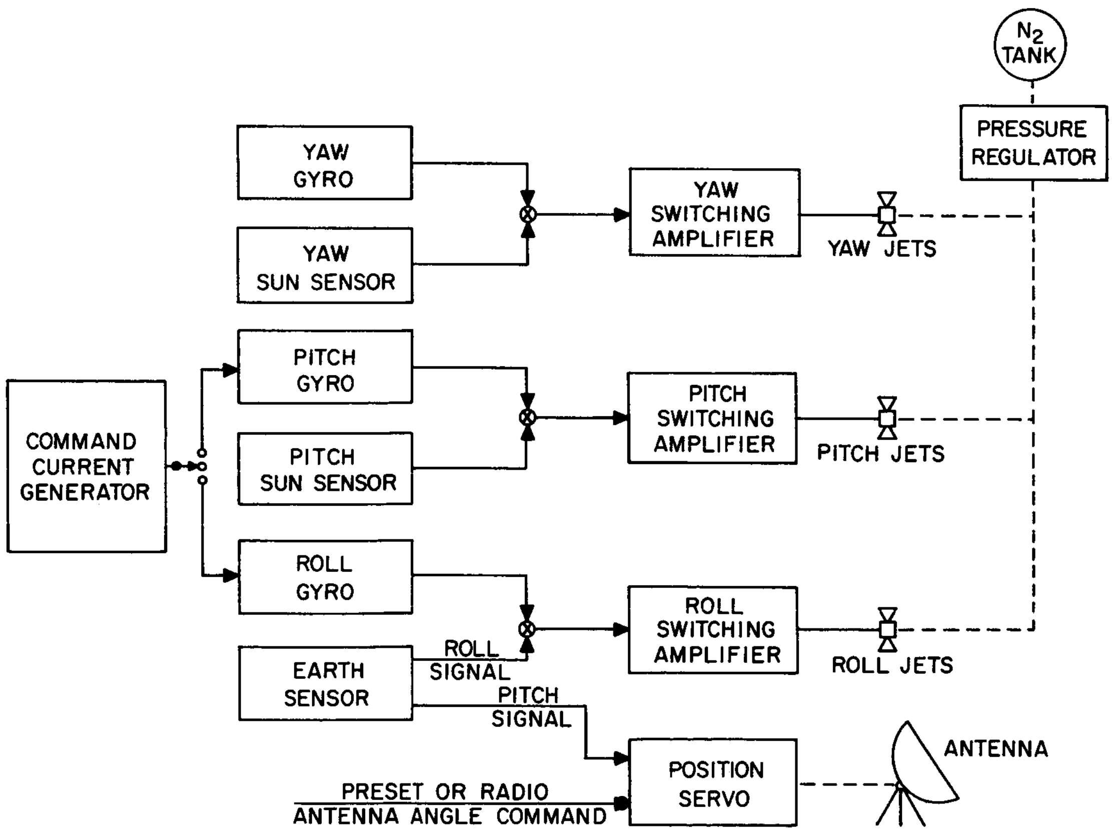

Fig. 15. Ranger attitude-control system